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6. PROPELLANTS

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INTRODUCTION

The mission that a rocket can accomplish is a function of the energy supplied to it and the weight of the vehicle. The performance of a rocket is often expressed in terms of the burnout or final velocity of the vehicle. This velocity is a function of the specific impulse of the propellant and the ratio of the gross weight to the empty weight of the vehicle.

This paper discusses propellants capable of achieving high specific impulse and some of the problems associated with their use. The following topics are of primary interest in this matter:

(1) propellants available and their specific impulses

(2) combustion efficiency

(3) combustion instabilities

(4) combustor cooling

A propellant combination is of little use unless it can be burned efficiently in an engine; this presents the problem of obtaining maximum combustion efficiency in as small an engine as possible. Efficiently burning high-energy propellants are used to propel extremely complex missile systems. The various missile and engine components have oscillatory modes which can oscillate in such a way that the oscillations reinforce each other and can even oscillate badly enough to tear the missile completely apart. Thus, combustion instability is of great interest.

High-energy propellants usually produce higher temperatures; the rates of heat transfer to the walls of the thrust chambers are going to be higher. Consequently, combustor cooling achieves greater importance.

Propellants

Propellant combinations that have a reasonable chance of being used in rocket propulsion are represented by those tabulated in table I, which

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gives the theoretical maximum sea-level specific impulses for a number of propellants. These propellants are divided into two groups: storable propellants, which require very little or no last minute preparation of the missile for launching, and nonstorable propellants, which require filling or topping off of the propellant tanks before the missile is launched. The propellants in the second group are nonstorable because they contain cryogenic liquids (i.e., liquefied gases).

The specific impulses are shown for combustion-chamber pressures of 300, 600, and 1000 pounds per square inch absolute. The storable liquid propellants represented by the RP-1 - red fuming nitric acid combination have higher specific impulses than the conventional solid propellants represented by the cast composite propellant.

The newer solids, represented by the polyurethane-aluminum-perchlorate and the polyvinyl chloride-diester-aluminum-perchlorate propellants, have impulses which reach into the regime between 260 and 270 and are competitive with storable liquid propellants. Solid-propellant rockets usually achieve some advantage in specific impulse by operating at higher combustion-chamber pressures than the liquid-propellant rockets. But they must pay a weight penalty for this increase in specific impulse in the form of larger, heavier thrust chambers which must be large enough to contain all the propellants and strong enough to withstand the higher pressures.

The most energetic propellant combinations are the exotic liquids, which are listed on the bottom part of table I. One combination, hydrazine with chlorine trifluoride, is storable. The remaining combinations are nonstorable because they contain the liquefied gases - fluorine, oxygen, or hydrogen. The most energetic of all these propellant combinations is the hydrogen-fluorine system which has a specific impulse of 409 at a combustion-chamber pressure of 1000 pounds per square inch absolute.

With liquid propellants, 409 is probably the highest possible specific impulse obtainable using conventional combustion reactions, unless liquid ozone is considered as a possible oxidizer. Hydrogen-ozone is expected to give a theoretical specific impulse of 386 at a combustion-chamber pressure of 300 pounds per square inch absolute. However, since this oxidizer is very difficult to handle and store, liquid ozone is not considered as a likely possibility at the present time. The cryogenic liquids now being considered give the highest specific impulses that can be reasonably expected with ordinary combustion reactions.

Solid propellants have been limited in the past by the stringent requirements of their physical properties. In addition, these propellants must be stable, unreactive, and storable. These requirements have given the solid propellants their well deserved reputation for reliability and "off the shelf" availability. Since the reactivity of the propellant usually goes up as the heat of reaction increases, solid propellants having high specific impulses have been difficult to make. However, it is reasonable to expect that solid propellants having high specific impulses will be developed, and novel packaging techniques will be perfected for using these propellants.

One approach to synthesizing reliable high-energy storable packages might be to combine the advantages of fuel-oxidant separation of liquid-propellant rockets with the convenience and reliability of fuel-oxidant storage in the thrust chamber of solid-propellant rockets. Reactive propellant combinations might be stored within the thrust chamber as they are in the solid rocket, but separated from each other, as they are in the liquid rocket, by relatively unreactive but combustible plastic sheets, tubes, or capsules instead of metal tanks. One technique for doing this is illustrated in figure 1.

For obvious reasons, the resulting grain (fig. 1) is called the candle type grain. It consists of a core of lithium perchlorate oxidizer that is protected from the fuel on the outside by a polyester-styrene copolymer; this inner core, in turn, is surrounded by the fuel, in this case lithium metal. Preliminary experiments at this laboratory have shown that this kind of grain will burn smoothly and vigorously and is safe and easy to handle. If new packaging techniques are considered, there is reason to believe that specific impulses higher than those tabulated can be achieved.

The effect of the heat of reaction on propellant performance is shown in figure 2 in which the theoretical specific impulse I_s is plotted against the heat of reaction. Data are presented for conventional double-base propellants, composite propellants, and a group of N-fluoro derivatives, which were proposed by Dr. Niederhauser of Rohm and Haas at the June, 1957 meeting of the joint Army-Navy-Air Force Solid Propellants Group. There is a great deal of scatter of the points because the specific impulse does not depend on the heat of reaction alone. It is a function of the molecular weights and heat capacities of the products as well. In general, as the heat of reaction goes up, the specific impulse of the propellants increases.

In order to illustrate future trends, vertical lines (fig. 2) indicate the heats of reaction for three of many other possible propellant combinations. These are a hydrocarbon containing 10 percent lithium as fuel with nitrosyl perchlorate as the oxidizer, decaborane with lithium perchlorate, and lithium with lithium perchlorate. Although these lines have been extended to intersect the curve, the point of intersection is not significant and should not be used to estimate specific impulse. The properties of the products of these reactions will undoubtedly be different from those of more conventional propellants. The molecular weights will be higher, and the specific impulses will not be as high as might be

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expected by extrapolating to the ordinate. However, figure 2 does indicate that there is a possibility of obtaining higher energy solid propellants.

Free radicals and the specific impulses obtainable from them are becoming of increasing interest in rocket propulsion work. Free radicals are fragments of molecules. They are obtained from ordinary molecules by breaking the chemical bonds in these molecules. In order to break the chemical bonds, energy must be supplied to the molecule, but this energy can be recovered when the fragments are either burned or allowed to recombine. For example, hydrogen molecules are composed of two hydrogen atoms that are held together by a very strong chemical bond. If enough energy is supplied, this bond can be broken and the atoms can be separated. For hydrogen, 93,000 Btu's are needed to separate 1 pound of hydrogen into its atoms. When these atoms are allowed to recombine, all this energy is liberated. In comparison, the heat of combustion of hydrogen with oxygen is only 6800 Btu's per pound.

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Very high specific impulses should therefore be obtainable from free radicals. But, unfortunately, the problem is not one of allowing the free radicals to recombine but to keep them from recombining until recombination is wanted. The Bureau of Standards is currently studying the fundamental chemistry and physics of free radicals. Included in their studies are attempts to isolate, stabilize, and concentrate free radicals. However, the possibilities of their use for rocket propulsion appear dim, as the highest concentration as yet reported of the free radicals which might be useful in rocket propulsion is about 1 percent. There are theoretical reasons for believing that the highest concentration capable of being stabilized will be about 16 percent.

Still, free radicals present an intriguing if remote possibility. This is illustrated in figure 3 where the specific impulse I_s of hydrogen atoms frozen in a hydrogen matrix at 0° R is plotted against atom concentration. The reaction products are expanded from a pressure of 300 pounds per square inch absolute in the combustion chamber to atmospheric pressure. The bottom curve assumes that thermodynamic equilibrium is achieved in the combustion chamber and that the gases are expanded in the frozen state through the nozzle. The middle curve assumes the same condition in the combustion chamber but assumes that thermodynamic equilibrium is maintained in the nozzle. The top curve uses the ordinate at the right and indicates the combustion-chamber temperature associated with this reaction.

The system of hydrogen atoms in a hydrogen matrix gives the highest specific impulse. If nitrogen atoms are substituted for hydrogen atoms, a maximum specific impulse of about 500 is reached at a nitrogen atom concentration of slightly over 20 percent. If imine radicals are substituted for the hydrogen atoms, a maximum specific impulse of about 450

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is obtained at about 45 percent. It might be expected that, as the nitrogen atom or imine radical concentration increases, the performance should increase. However, the substitution of the heavy nitrogen atom for a light hydrogen atom increases the molecular weight of the product gases so that the specific impulse actually reaches a maximum at the points shown (fig. 3) and then begins to decrease again.

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Combustion Efficiency

An idealized rocket model is shown in figure 4 in order to explain the important concepts in the combustion of rocket propellants. The oxidant and fuel are injected into the combustion chamber through holes in the injector. This figure shows two propellant streams impinging upon each other, a characteristic of a like-on-like injector. After some time and distance, the propellants are atomized into oxidant and fuel drops which vaporize as they move down the combustion chamber. Since there are both large and small drops, the rate of vaporization will vary both between drops and with distance. As the propellants vaporize, they mix and then react to form the desired hot combustion gases.

Considering atomization, mixing, vaporization, and reaction and their dependence on various design and operating parameters produces a very complex problem. One might expect that this problem might be simplified by isolation of the process that requires the greatest distance.

The rocket engine can be compared with the more familiar ramjet. The three significant differences between these two propulsion systems are shown in the following table:

Operating conditions	Rocket	Ramjet
Combustion-chamber pressure, atm	40	1/2
Propellant concentration, percent		
Fuel	20	3
Oxidant	80	20
Cross-sectional area, sq in./(lb/sec)	1	1000

The rocket operates at high combustion pressures, but the ramjet functions at a very low pressure. The propellant concentration of the ramjet is considerably lower than that for the rocket. A great amount of liquid propellant must be burned in a very small cross-sectional area in the rocket because its area is so much smaller than that of the ramjet.

These operating conditions affect the time and distance required for the processes that take place in the combustion chamber. This distance is analyzed in the following table:

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Process	Relative distances	
	Rocket	Ramjet
Atomization	1	1
Vaporization	30	5
Mixing	1	4
Chemical reaction	<1	20

Since the atomization distance is about the same in both systems, it has been assigned a relative value of 1 for both systems. The lengths for the other processes are therefore relative to the distance required for atomization. 4793-I

The length necessary to vaporize the propellants in the rocket engine is about six times that of the ramjet. This increase in length is due to the larger drops produced in the rocket engine. Mean drop sizes in a rocket engine are about 200 microns, whereas in the ramjet the drop size is in the order of 50 microns.

The slightly lower mixing length in the rocket engine is due to a very low cross-sectional area in the rocket. As a result, the distance between the streams which control the mixing distance is reduced.

The chemical reaction process requires a distance of <1 in the rocket engine as compared with 20 in the ramjet. This greatly reduced length occurs because the reaction distance decreases as the temperature is increased and the concentration of the propellant or the pressure increases. The increase in pressure in the rocket decreases the reaction length by a factor of 80. In addition, the increase in concentration and temperature will also reduce the length.

The previous table indicates that in the rocket engine the greatest distance is for vaporization, while the distances for atomization, mixing, and chemical reaction are all small in comparison. Thus, the distances required for atomization, mixing, and chemical reaction can be neglected, and the combustion efficiency in a rocket engine is assumed to be proportional to the fraction of the total propellant that has been vaporized in the engine.

The way in which the various parameters affect combustion efficiency should be considered. Varying the length of the combustion chamber is the first thing to consider. By using known vaporization equations the percent of propellant vaporized can be calculated as a function of chamber length as shown in figure 5. The engine efficiency is approximately equal to the percent of propellant vaporized. The slope of the curve for the calculated results continually decreases because the drops get smaller and smaller as they vaporize. The curve asymptotically approaches 100 since the largest drop is never completely vaporized.

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The experimental results for a hydrogen-oxygen engine using the same injector agree well with the analytical curve.

The curves for all the propellants are characterized by two properties: where the knee occurs and where high efficiency is obtained. In the following discussion the length for 90-percent efficiency will be used to represent where the knee occurs in the curve, and the length for 99-percent efficiency will be used to show where high efficiency is obtained.

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Since vaporization is the important parameter in the combustion of rocket propellants because of larger drop sizes, the drop size can be reduced in order to improve the efficiency. One of various fundamental concepts that can be used to reduce drop size is orifice diameter. The results of this test are shown in figure 6. Drop diameter is plotted against orifice diameter for an injection system similar to the like-on-like injector where two streams impinge upon each other. Gas was introduced behind the liquid stream to carry the drops down the chamber. Drop size reduced as the orifice diameter was decreased. The curve ends at an orifice diameter of 0.020 inch as this is the practical minimum that can be used in a rocket engine without the injector becoming plugged by foreign particles. Increasing the velocity difference, that is, the difference between the gas and liquid stream velocities also decreases the drop size.

The two chamber lengths defining the efficiency curve (i.e., those for 99- and 90-percent efficiencies) are plotted against injector orifice diameter in figure 7. The results of the analytical study, which calculates the percentage of propellant vaporized, are shown by the solid line. The experimental results for a JP-4 - liquid-oxygen system are shown by the squares. The length required for high efficiency decreases as the injector orifice size is decreased. The slope of the experimental results is even greater than that predicted analytically, which indicates that more is gained by decreasing injector orifice size than predicted. The results for the hydrogen with liquid-oxygen system, represented by the triangles, show the same characteristics.

Another parameter that affects engine efficiency is the difference in velocity between the drops and the combustion gases. Increasing this velocity difference increases the heat-transfer rates to the drop by reducing the boundary layer around the droplet. Increasing the gas velocity is the easiest way to increase the difference between the velocities of the gases and the drops; this can be done by building engines with lower chamber-to-throat diameter ratios.

The importance of slimmer engines on the combustion-chamber length required for high efficiency is shown in figure 8. A ratio of chamber-to-throat diameter of 3, compared with a ratio of 1, has resulted analytically in a reduction of one-half the chamber length required for high efficiency. Since the slope of the curve through the experimental values for a JP-4 - liquid-oxygen engine using a like-on-like injector is greater than the slope through the analytical calculations, the actual effect of diameter ratio is even greater than had been predicted.

A comparison of the experimental results obtained using the analytical model with various propellants is presented in table II. These experiments were all done in a 200-pound-thrust engine, and the numbers represented in the table are the average of 15 or more tests. The same injector, chamber diameter, and liquid-propellant orifice size were used for all the tests. The ammonia and JP-4 fuels, both with liquid oxygen as the oxidizer, required 60- and 48-inch distances, respectively, in order to achieve 99-percent efficiency. Liquid ammonia and hydrazine, using liquid fluorine as oxidizer, required lengths of 65 and 56 inches, respectively. These four propellants are characterized by having fuels with high boiling points relative to the boiling points of the oxidizers, oxygen and fluorine. When gaseous hydrogen was used as a fuel with the cryogenic oxidizers, the lengths required to achieve 99-percent efficiency were reduced to 17 and 19 inches. This indicates that cryogenic fuels with cryogenic oxidizers vaporize faster and give higher combustion efficiencies in shorter thrust chambers.

Since drop size is a function of the injection velocity difference, another way of improving combustion efficiency is to decrease the drop size by having a high gas flow behind the liquid-propellant stream. The results of this test are plotted in figure 9 in which chamber lengths for 90- and 99-percent efficiencies are plotted against the injector pressure drop. The higher the pressure drop, the higher will be the velocity of the hydrogen. The injector with a high pressure drop has a very small orifice behind the liquid stream; for a low pressure drop a large-diameter orifice is needed behind the liquid stream. Increasing the gas velocity by increasing the pressure drop resulted in a shorter length for high efficiency. This indicates that a high hydrogen pressure drop is beneficial.

A model has been described which uses vaporization as the rate-determining process in establishing combustion efficiency and which gives results that are consistent with experimental data. This suggests that engines, in order to achieve maximum efficiency, should have thin chambers and small holes in the injector. Although these conditions are important in achieving the desired efficient combustion, the complete system should maintain its mechanical stability.

Combustion Instabilities

Figure 10 illustrates the flexible, elastic nature of a missile, which is represented as four mass concentrations - the motor, the oxidant

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tank, the fuel tank, and the nosecone. The springs represent the flexible framework between the masses; the tank bottoms, cooling passage walls, and injector faces are also shown as flexible surfaces.

There are many ways in which such a structure can oscillate or vibrate. For example, aerodynamic loading at the nosecone can introduce disturbances which oscillate the propellant tanks and lines, thereby affecting the flow of propellants to the motor. The thrust produced by the motor can similarly cause pulsations or oscillations in propellant flow. Variations in combustion-chamber pressure also affect the flow into the motor by deflecting the cooling passage walls and by directly affecting the pressure drop across the injector.

Disturbances such as those produced, for example, by drag or thrust forces do not determine whether the system is stable or unstable. These forces only shake or vibrate the system, and this is not an instability in itself.

On the other hand, several paths have been indicated through which the combustion-chamber pressure can affect the flow. Since flow in turn affects the chamber pressure, a situation is present where even a very small disturbance can be perpetuated and amplified if conditions are right. For example, consider the effect of combustion-chamber pressure on flow through the cooling passages. Chamber pressure affects propellant flow, flow affects pressure, the new pressure affects flow, and so forth. Thus, an unstable system drives itself. In general, the oscillations that are generated build up until the system oscillates violently and sometimes destroys itself.

At this point it will be helpful to examine one of these feedbacks more closely. Figure 11 shows a sketch of a motor, an injector, and a tank supplying propellant to the injector at a constant pressure. The important elements which determine whether oscillations, chugging, will occur in this system are the variation of propellant flow with pressure drop, the variation of combustion-chamber pressure with propellant flow, and the time constants associated with these processes.

The time constants are related to the motor response. If the flow to the motor is suddenly changed, as indicated in figure 11, there is no immediate change in combustion-chamber pressure p_c. After a period of time α, the combustion-chamber pressure will rise. As propellant is added to the chamber at the new flow rate, the chamber pressure will continue to rise. The time between the start of the pressure rise and the attainment of the new equilibrium value is represented by τ, the time for this portion of the response to be 63 percent completed.

Two factors are important in determining if a system will oscillate. The first is the sensitivity of the system, which is called the gain or

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amplification factor, and the second is the response of the system, which determines the conditions under which a reinforced signal is obtained. The amplification factor is related to $\Delta p/p_c$, which is the ratio of the injector pressure drop to the combustion-chamber pressure, and the response is related to α and τ.

A stability diagram for the engine-injector loop is presented in figure 12. The curve defines the regions of stable and unstable operation in terms of $\Delta p/p_c$ and α/τ. High ratios of $\Delta p/p_c$ and low ratios of α/τ improve stability. Low values of α are associated with short atomization, vaporization, and mixing times. These were previously shown to improve efficiency; they are shown here to improve stability. The time constant τ is proportional to the ratio of the chamber area to the throat area. Since low area ratios improve efficiency, good stability, therefore, may not always be compatible with good efficiency.

As shown in figure 12, stability can always be achieved by increasing the pressure drop across the injector sufficiently. If it is undesirable from the standpoint of the engine performance to do this, the same result can be achieved by inserting a restriction in the line ahead of the injector. However, any increase in pressure drop necessitates increased weights of the feed system in order to withstand the higher pressures.

Chugging stabilizers have been proposed also. Although these devices add complexity to the system, they do offer the hope of being able to operate at low values of injector pressure drop.

As previously indicated, there are a number of feedback paths to be considered in a missile system. Figure 13 presents some of the results of a study in which only one feedback is included through the framework. The system is represented as one mass for the upper portion of the missle, a flexible framework for the motor supports, and another mass for the motor.

At low time-constant ratios the stable range is decreased because of the flexibility, but for high values of α/τ the stable range is increased. The shape of the curve and the range of stable operation available depend on the masses, spring rates, fluid inertia, and compressibility effects, among other things. With many feedback paths and under other operating conditions the stable range can be changed considerably more than indicated here.

Figure 13 indicates that a larger stable range was available with the rigid system than with the original engine-injector system alone. This increase in stable operating range is the result of including propellant line and pump dynamics.

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Another form of oscillation that rocket engines are subject to is the oscillation of the gases in the combustion chamber or "screaming." Screaming is associated with a pressure wave that travels back and forth in the combustion chamber and leads to high local rates of chemical reaction, high rates of heat transfer, and hot spots in the engine. Such hot spots can cause an engine to burn out in less than 1 second.

Screaming usually consists of oscillations in one of two modes. The first, a longitudinal mode, travels from the nozzle to the injector, reflects, and then travels back to the nozzle. The frequency of this wave is associated with the length of the chamber. The second is a transverse mode in which the wave travels around or across the chamber. The frequencies in this wave are associated with the diameter or the circumference of the chamber. Harmonics of either of these types of waves can also occur and have been observed experimentally.

The type of wave obtained is dependent on the geometry of the chamber as illustrated in figure 14. With a longitudinal mode the energy dissipated decreases as the length-to-diameter ratio increases. For the rotary mode (a transverse mode), the energy required is independent of the length-to-diameter ratio. Since the wave requiring the lowest energy will prevail, a large length-to-diameter ratio engine should give the longitudinal wave, and the low length-to-diameter ratio engine should have the rotary wave. For engines having length-to-diameter ratios in the region of 5, either wave may be expected as the energy required is about the same for either wave.

The mechanism for sustaining a pressure wave was postulated in 1877 by Rayleigh who said: "If energy is added to the gas at the moment of greatest pressure, or absorbed at the moment of lowest pressure the vibration is encouraged". The rate of energy addition to the wave can be perturbed by at least two factors: (1) a change in chemical reaction rate,

$$r = P^m e^{-K/T}$$

where

r rate

P pressure

m constant

K constant

T temperature

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and (2) a change in vaporization rate,

$$r \approx \left(\frac{V_{diff}P}{\mu T}\right)^{1/2} \frac{\Delta T}{D_{drop}^{3/2}}$$

where

V_{diff} velocity difference between drops and gas

μ viscosity of gases

ΔT temperature difference between drops and gas

D_{drop} diameter of drops

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The most obvious method of adding energy to a wave is by a perturbation in the chemical reaction rate. The chemical reaction rate is dependent on the pressure at which the reaction is occurring and the temperature of the reaction. Increases in either pressure or temperature will increase the reaction rate. Thus, the perturbation in energy released is in phase with the pressure and temperature waves.

Vaporization rates should also be considered since vaporization is very important in rocket-engine combustion. The vaporization rate is proportional to the velocity difference between the gas and the drop, the pressure of the system, and the temperature difference between the drops and the surrounding gas. Increasing the velocity difference, pressure, or gas temperature will increase the reaction rate.

Increased pressure leads to increased temperature, and increased temperature gives increased reaction rates. A pressure wave should be accompanied by a velocity wave that is 90° out of phase with it; the velocity change will affect the evaporation rate. These waves have been observed experimentally in a rocket engine, and the results are shown in figure 15, which shows the pressure, temperature, and velocity histories inside the combustion chamber. The temperature is in phase with the pressure; the velocity is 90° out of phase with the pressure.

Screaming is related to the energy release under steady-state conditions as is shown in figure 16. The solid symbols represent longitudinal and rotary screaming, and the amount of shading is characteristic of the fraction of the runs that screamed. For low-energy propellants, a system having a rapid conversion rate of the propellants to hot gases and a performance curve in the shaded area was inherently unstable. A system having a lower conversion rate and producing a performance curve in the unshaded region was stable.

This same phenomenon has been observed for the high-energy propellants. An engine with a rapid conversion rate was inherently unstable; however, with a lower conversion rate the system was stable. One interesting difference between these two systems is that the high-energy propellant system with the rapid rate of conversion and high energy release reduced the region in which screaming was obtained. This can be observed by comparing the shaded areas for the high- and low-energy systems.

The screaming region can also be reduced by increasing the damping in the system. Screech in turbojet afterburners was eliminated by inserting perforated liners to introduce damping. A similar approach was used in rockets by placing baffles in the combustion chamber. The results of this investigation are shown in figure 17. Without baffles, 78 percent of the tests with the engine were screaming runs. With baffles, the percent of the screaming runs was reduced to 5 percent.

Screaming is a major problem because it is accompanied by an increased heat-transfer rate which burns out the engine. In addition, during screaming the combustion process is changed. A localized increase in combustion rate seems to produce localized hot spots next to the injector, which can also cause the engine to burn out.

Cooling

Although the higher energy propellants produce higher heat-transfer rates, these propellants can absorb greater quantities of heat when they are used for cooling. The simplest and cleanest way to cool a rocket engine is to pass one of the propellants through the cooling passages before it goes to the combustion chamber. This process is called regenerative cooling.

The cooling capacities for several rocket propellant combinations are presented in figure 18. The engines are assumed to operate at the oxidant-fuel ratio of maximum specific impulse and at a combustion pressure of 300 pounds per square inch absolute. The shaded bars represent the fuels, and the open bars represent the oxidants.

Generally, the limitations of cooling capacities are brought about by the physical properties of the fluid itself. For example, jet fuel, ammonia, and hydrazine are limited by their boiling points at the pressures in the cooling jacket. Hydrazine is limited further by the fact that it decomposes thermally at temperatures near this boiling point. Hydrogen, however, has no limitations due to physical properties because it is considered to be above its critical pressure. This means that there will be no phase transition, or no boiling. The limit for hydrogen cooling is imposed by the metal of the engine walls, which cannot be heated above the limits tolerable for structural integrity. Two values are shown for

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hydrogen on figure 18 because it is used with both fluorine and oxygen as oxidants. When hydrogen is burned with oxygen, more fuel is needed, and thus a greater cooling capacity is available.

The use of oxidants as coolants presents some promise. However, as with hydrogen, liquefied gases are being considered. Since the critical points for oxygen and fluorine are somewhat high, a choice must be made as to whether or not the coolant is to be used above or below critical pressure. If the coolant is used below the critical pressure, it is limited by the boiling point. The solid portions of the oxidant bars (fig. 18) represent the heat capacities available within the limitations of the boiling points of the fluid at pressures normal for cooling. However, if higher pressures are used, for example 800 pounds per square inch, then the critical pressure is exceeded and there is no boiling point problem. Again, the engine wall provides the limit. The total heat capacity is represented by the total height of the bar for each oxidant.

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The cooling capacities of various propellants have been discussed; next, the cooling needs of rocket engines must be considered. Figure 19 presents the results of an analysis of the cooling requirements for 10,000-, 100,000-, and 1,000,000-pound-thrust engines in terms of the ratio of the cooling required by the engine to the cooling available from the fuel.

Regenerative cooling would not be possible above a cooling ratio of 1. The engines of this investigation were assumed to operate at the oxidant-fuel ratio of maximum specific impulse at a combustion pressure of 300 pounds per square inch absolute. The fuel alone was considered as the coolant, and the cooling process was purely regenerative. In all cases, increasing the thrust level decreases the heat load on the coolant because, as the engine is increased in size, the surface area does not increase as fast as the volume.

For the propellant combinations containing hydrogen, the total cooling capacity required of the hydrogen, by analysis, does not reach the limit of that available. The adequate cooling capacity of hydrogen has been demonstrated experimentally by the NACA.

When JP-4 fuel with liquid oxygen is used, a carbon film is deposited on the gas side of the cooled wall as a self-renewing insulator. This carbon film was taken into account in these analyses. Experimental data are plotted for the JP-4 - liquid-oxygen combination for 1000- and 5000-pound-thrust NACA engines and for Rocketdyne sustainer and booster engines for ICBM use. Rocketdyne personnel have evidence which leads them to believe that the carbon layer builds up on the engine wall, flakes off, and rebuilds again. Thus, the local heat-transfer rate would be transient and possibly cyclic in nature. The net effect of the carbon film is to reduce heat transfer. The experimental data and the analysis for this case are in reasonable agreement.

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For the hydrazine-fluorine combination, cooling of at least the larger engines appears to present no problem if thermal decomposition can be avoided. It is assumed that the stay time of the hydrazine in the coolant passages will be short and that the velocity of flow through the passages will be fast enough that the fluid will reach the combustion chamber before it decomposes. If decomposition occurs with hydrazine, it becomes a very good monopropellant. This sort of behavior is not wanted in the cooling jacket.

Cooling with ammonia appears to be quite marginal, at least with smaller engines. NACA data obtained at 1000-pound thrust with a water-cooled engine indicate, however, that purely regenerative cooling might even be possible with these small engines.

In experimental work, heat rejection rates are usually obtained which are only about 60 percent of those calculated. In any such calculations, certain assumptions must be made. For the present analysis, the assumptions were conservative. For example, combustion has been assumed to be perfect, which means that full combustion temperature was reached, less fuel was available because of higher performance, and hence less coolant was available. It was also assumed that the gases in the chamber would have homogeneous distribution and that the temperature along the walls was uniform, even back to the injector face. In addition, any effects of the injector such as hot and cool spots, which may be functions of the propellant distribution, were not considered.

The experimental data at 3000-pound thrust in figure 19 are from Rocketdyne for ammonia-fluorine. All measurements were made with engines of the same design, but the injectors were varied. The highest heat-transfer rate was obtained with a doublet-type injector, that is, each fuel jet impinged on an oxidant jet; the lowest heat-transfer rates were obtained with a like-on-like injector. The intermediate point was obtained with a hybrid of these two injector types. All the experimental data points in figure 19 were adjusted to conform to the operating conditions assumed in the analysis.

Data from Bell Aircraft show that cooling may be accomplished with ammonia in engines big enough to be used as sustainers. The heat-transfer rate, however, for these engines was about 5.6 Btu/sec-sq in. as compared with 1.1 for a corresponding sustainer engine using jet fuel and liquid oxygen.

Because of this high heat-transfer rate and the questionable capability of cooling with ammonia, the addition of a ceramic lining inside the wall has been considered for ammonia-fluorine engines. In this analysis, the ceramic reduced the heat-transfer rate appreciably. Bell Aircraft recently experimented with a ceramic liner in an ammonia-fluorine engine. The liner appeared to reduce the heat flux, but it eroded very rapidly.

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Most refractories are oxides. The resistance of such refractories to attack by hot, turbulent fluorine gas is not known. Perhaps future research will result in fluoride-type refractories which will resist fluorine attack.

Cooling requirements are also influenced by engine parameters other than thrust level, oxidant-fuel ratio, and injection pattern. For example, at a given thrust level, decreasing the chamber diameter increases the gas velocity and slightly increases the heat load. Increasing the expansion ratio of the nozzle for high-altitude operation also increases the cooling demands because a bigger nozzle must be cooled. If the combustion pressure is increased, the heat load is also increased because of higher combustion temperatures, different transport properties, and higher mass-flow rates of combustion gases across the cooled wall. 4379-I

Because the hydrogen-fluorine propellant combination offers the highest performance potential of any stable chemical system, further consideration will be given to this system and the cooling problems associated with it. The data in figure 19 were for 15 percent hydrogen, which gives the highest specific impulse. Missile designers, however, would prefer less hydrogen, since the low density of hydrogen appreciately increases the weights of the tanks and the pumps. Figure 20 shows the variation of the cooling requirements with various proportions of hydrogen and fluorine. While the change in specific impulse brought about by decreasing the percentage of hydrogen is not very significant, decreasing the fraction of hydrogen from 15 to 10 percent doubles the cooling load. Decreasing it to 5 percent, the stoichiometric ratio, almost requires the complete cooling potential of the hydrogen because the flame temperature is higher and only one-third as much fuel can be used for cooling.

Even though hydrogen has been shown generally to have ample cooling capacity, it is not known whether this can be actually realized in practice. The mechanism of heat transfer through the hydrogen coolant film remains a research problem. Not enough is known about the conditions in this film to which transport data are applied.

Figure 21 presents some heat flux rates and coolant velocities as functions of cooled length in typical engines for hydrogen-fluorine and JP-4 - oxygen propellant combinations. The coolant enters at the end of the nozzle and flows toward the injector. The heat flux rate at the nozzle throat for hydrogen (near 10 Btu/sec-sq in.) is about four times that encountered in present engines. The velocity of hydrogen in the coolant passages is an order of magnitude higher than that of JP-4 fuel, a condition never before experienced. Exploratory analysis such as this is valuable, but experimental work is needed to solve the cooling problems.

To determine experimentally whether hydrogen-fluorine engines running at high efficiency can be cooled with hydrogen, a 5000-pound-thrust

[redacted]

engine that operates at a combustion pressure of 300 pounds per square inch absolute has been designed and built at this laboratory. It was formed of nickel channels with 0.020-inch-thick walls; the channels were wrapped with wire and brazed. The wire takes the combustion-chamber pressure load.

This engine, which was run successfully, gave high performance and ample cooling. A specific impulse of 351 was obtained at 18.9 percent fuel. The chamber pressure (nominally 300) was 380 pounds per square inch; the thrust was 5980 pounds.

Figure 22 shows design temperature and pressure profiles as functions of the engine length. Experimental measurements at terminal points are represented by the circles. The actual rise in coolant temperature was somewhat lower than calculated; the actual pressure drop also was lower. The experimental heat flux rate of about 5 can be compared with the analytical value of 6.88 Btu/sec-sq in.

A small engine has been regeneratively cooled successfully; bigger engines should prove easier.

[redacted]

THEORETICAL MAXIMUM SPECIFIC IMPULSE
EXPANDED TO SEA LEVEL ATMOSPHERIC PRESSURE

CHAMBER PRESSURE, PSIA	300	600	1000
PROPELLANT	MAXIMUM SPECIFIC IMPULSE $\frac{\text{LB. THRUST}}{\text{LB/SEC}}$		
STORABLE			
CAST COMPOSITE	193	208	217
RP1 - RFNA	227	248	264
POLYURETHANE-ALUMINUM-PERCHLORATE	228	253	261
POLYVINYL CHLORIDE-DIESTER-ALUMINUM-PERCHLORATE	234	253	265
HYDRAZINE-CHLORINE TRIFLUORIDE	261	281	295
NON-STORABLE			
RP1 - OXYGEN	261	285	299
AMMONIA - FLUORINE	312	338	355
HYDRAZINE-FLUORINE	315	344	361
HYDROGEN-OXYGEN	348	374	390
HYDROGEN - FLUORINE	365	393	409

CS-14880

Table I

CHAMBER LENGTH FOR VARIOUS PROPELLANTS

PROPELLANT				LENGTH TO OBTAIN (IN.)			
FUEL	B P, °F	OXIDANT	B P, °F	99% EFF		90% EFF	
				EXP	THEOR	EXP	THEOR
NH_3	-28	LIQ O_2	-297	60	56	22	21
JP-4	209	LIQ O_2	-297	48	50	18	17
NH_3	-28	LIQ F_2	-305	65	56	24	21
N_2H_4	236	LIQ F_2	-305	56	71	21	26
GASEOUS H_2	-423	LIQ O_2	-297	17	21	6	8
GASEOUS H_2	-423	LIQ F_2	-305	19	20	7	8

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Table II

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CANDLE TYPE GRAIN

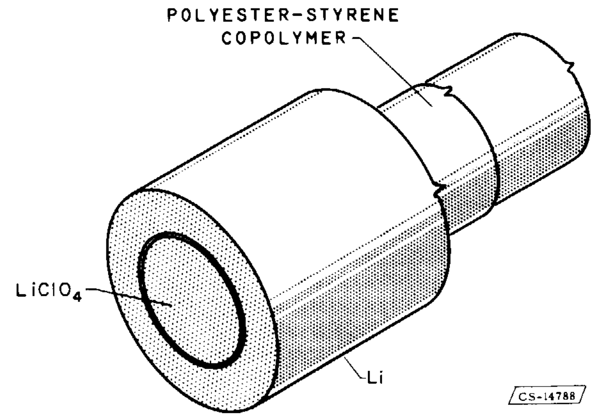

Figure 1

EFFECT OF HEAT OF REACTION ON PROPELLANT PERFORMANCE

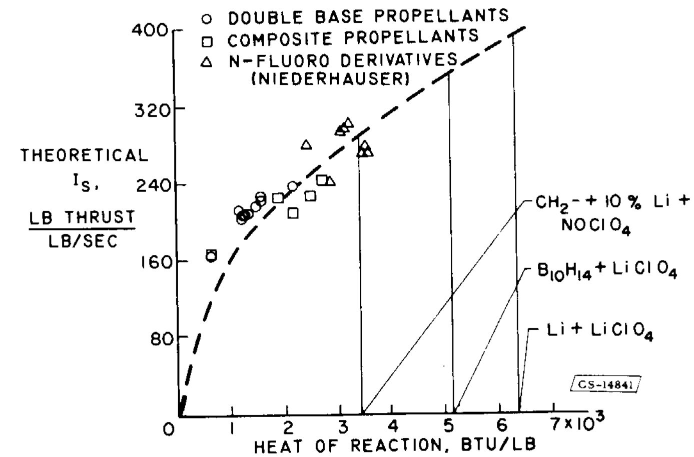

Figure 2

SPECIFIC IMPULSE OF FREE RADICALS

FROZEN AT 0° R

EXPANDED FROM 300 PSIA TO 14.7 PSIA

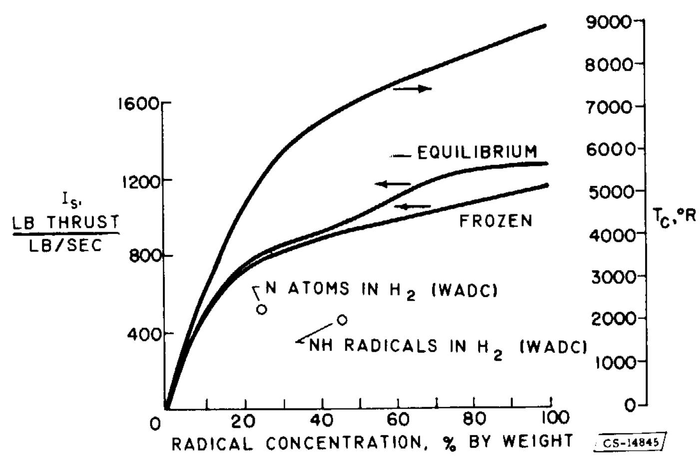

Figure 3

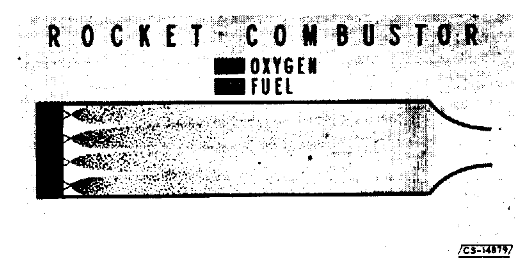

Figure 4

EFFECT OF CHAMBER LENGTH ON ENGINE PERFORMANCE

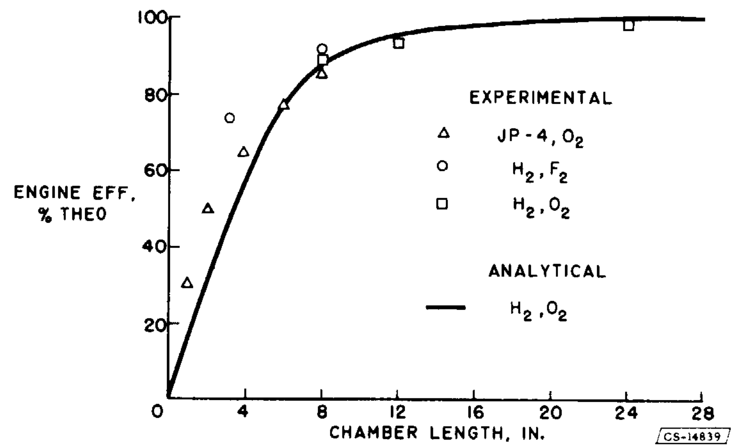

Figure 5

EFFECT OF ORIFICE DIAMETER ON DROP DIAMETER

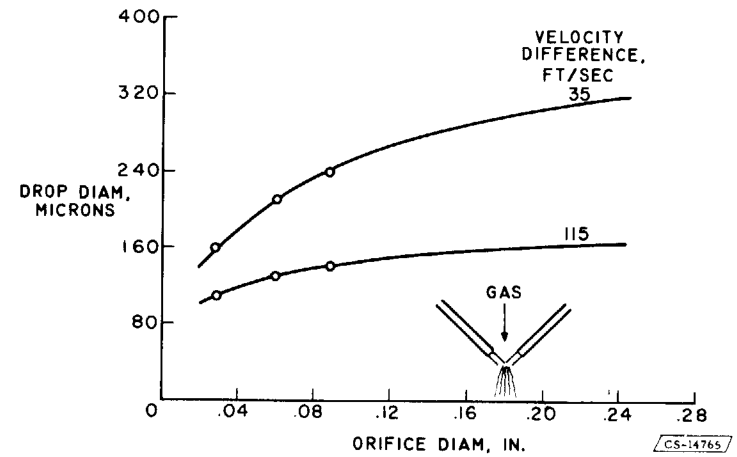

Figure 6

INJECTOR HOLE SIZE

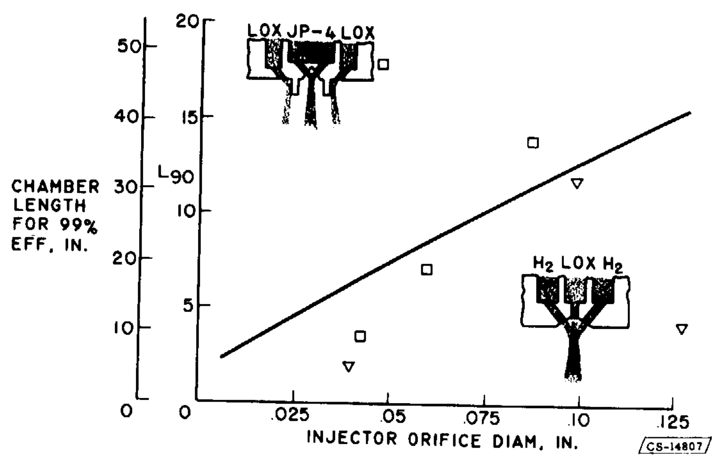

Figure 7

EFFECT OF CHAMBER-TO-THROAT DIAMETER RATIO

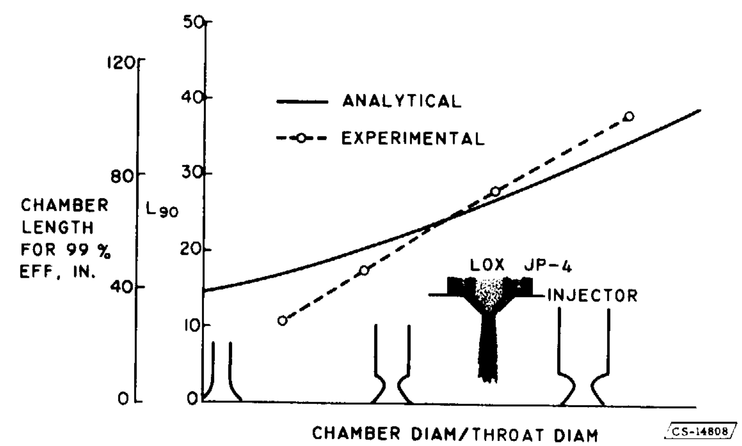

Figure 8

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EFFECT OF HYDROGEN PRESSURE DROP

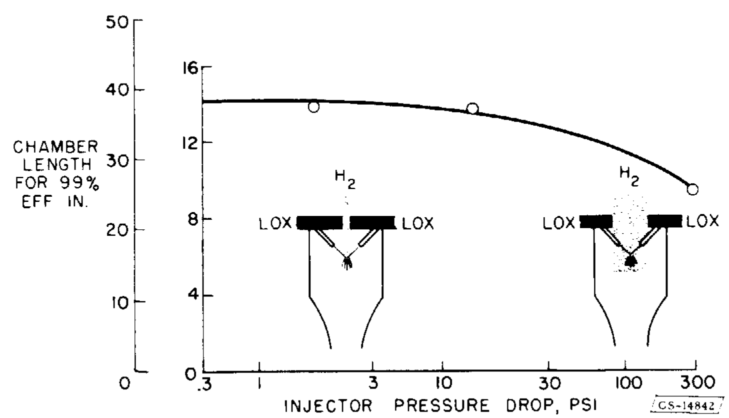

Figure 9

MODEL OF MISSILE DYNAMICS

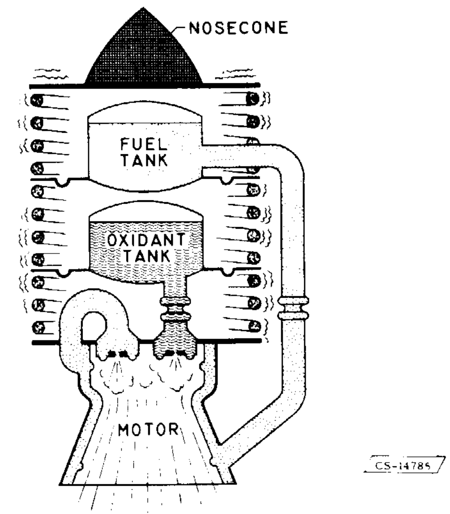

Figure 10

SCHEMATIC SHOWING INTERACTION BETWEEN MOTOR AND INJECTOR

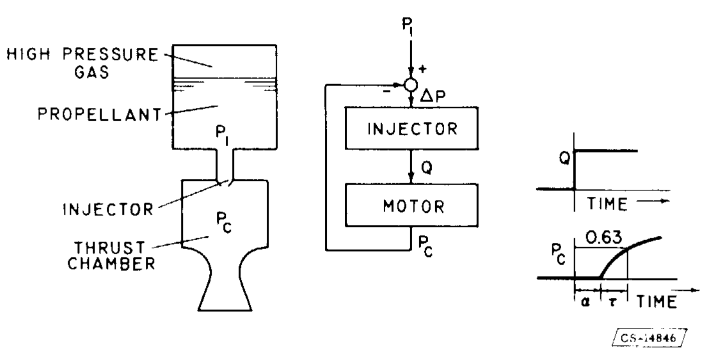

Figure 11

STABILITY DIAGRAM FOR ENGINE-INJECTOR LOOP

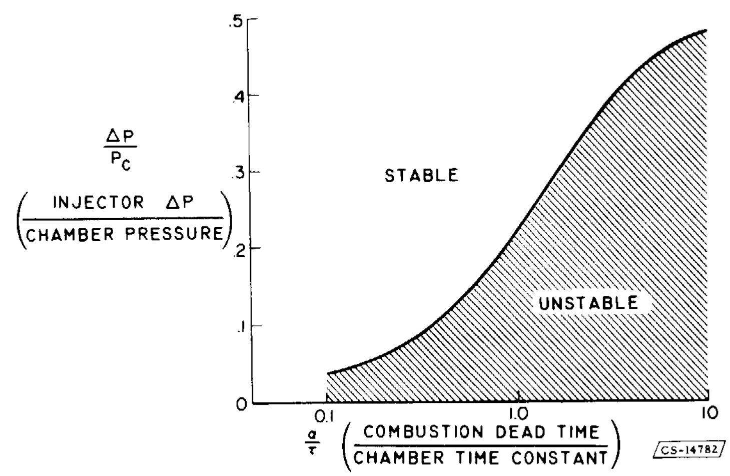

Figure 12

EFFECT OF FRAME FLEXURE ON SYSTEM STABILITY

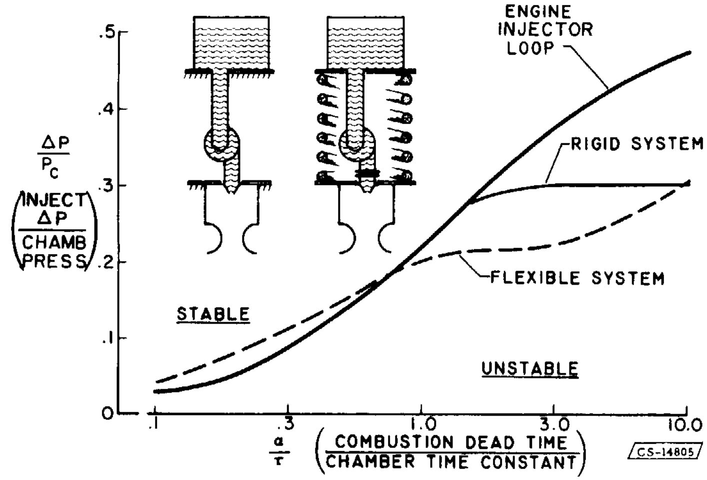

Figure 13

ENERGY DISSIPATION DURING SCREAMING

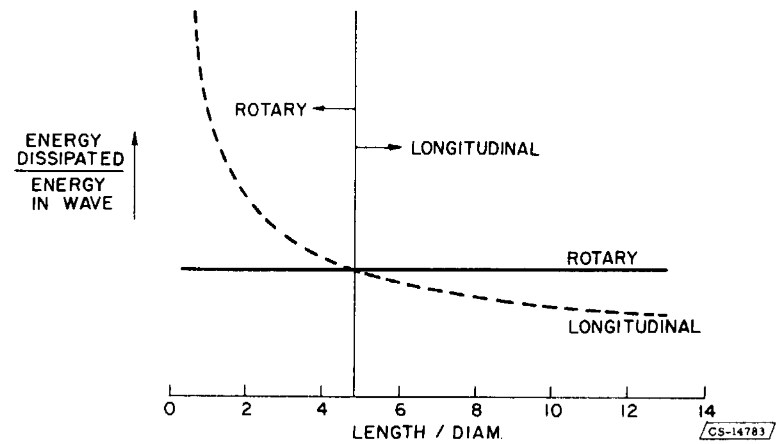

Figure 14

PRESSURE, TEMPERATURE, AND VELOCITY HISTORIES

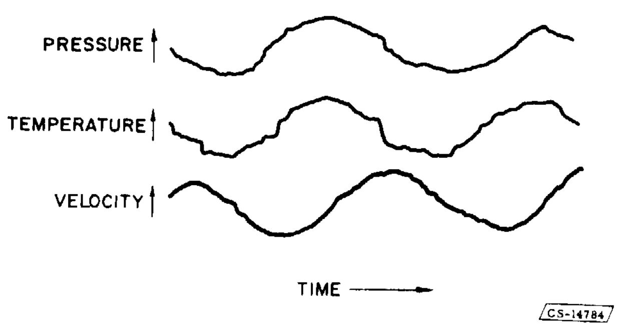

Figure 15

SCREAMING REGION

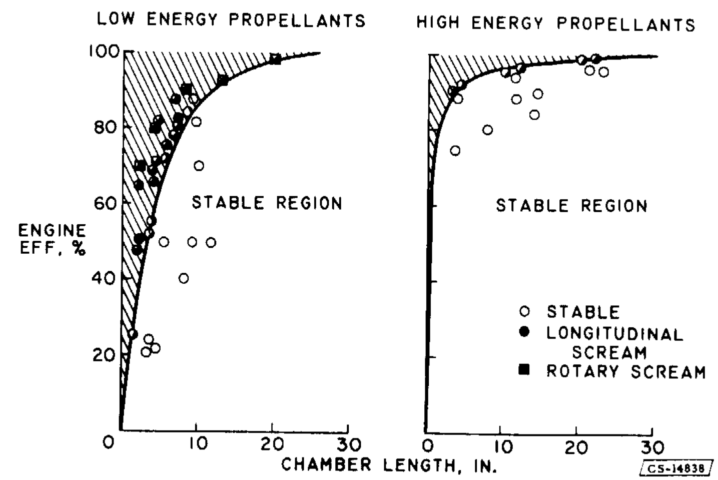

Figure 16

USE OF BAFFLES TO PREVENT ROTARY SCREAMING

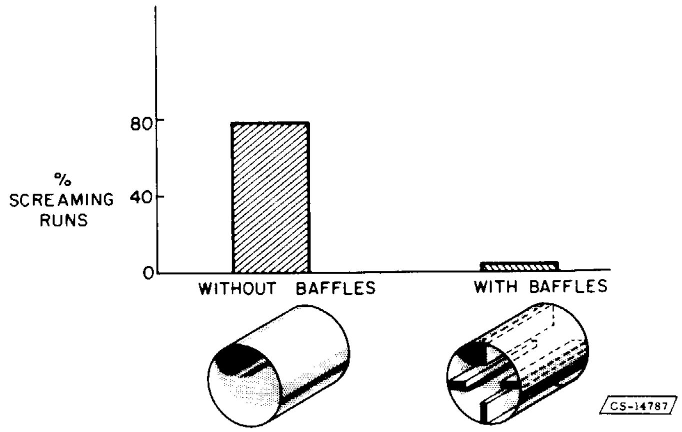

Figure 17

COOLING CAPACITIES OF PROPELLANTS

CHAMBER PRESSURE, 300 PSIA MAXIMUM SPECIFIC IMPULSE

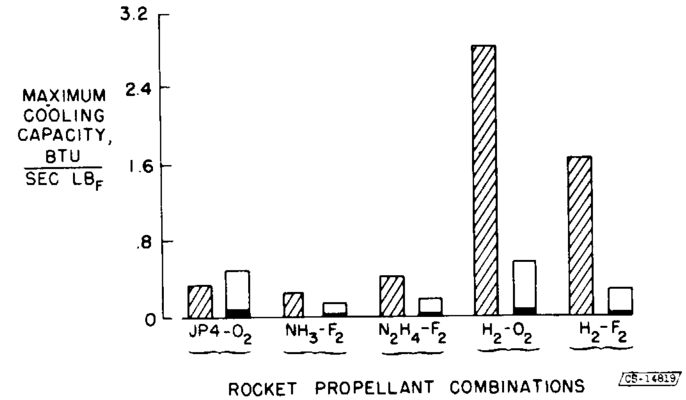

Figure 18

ENGINE COOLING POTENTIAL OF ROCKET FUELS

CHAMBER PRESSURE, 300 PSIA MAXIMUM SPECIFIC IMPULSE

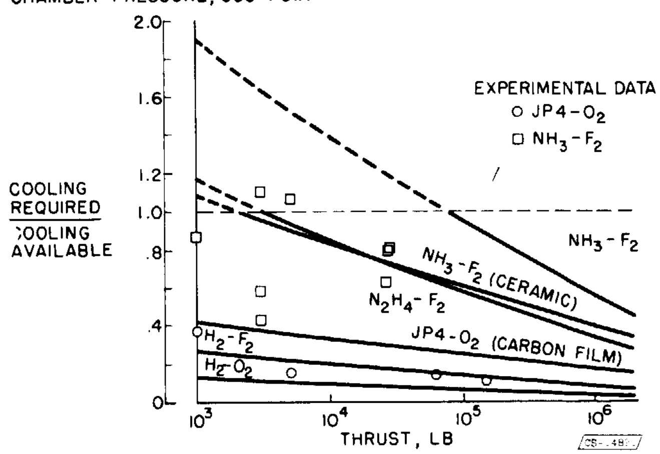

Figure 19

COOLING POTENTIAL FOR H_2-F_2 ENGINES

CHAMBER PRESSURE, 300 PSIA THRUST, 100,000 LB

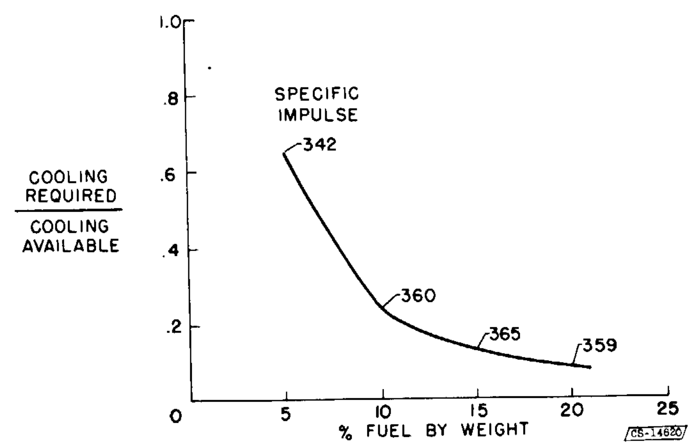

Figure 20

[redacted]

ROCKET ENGINE HEAT FLUX AND COOLANT VELOCITY

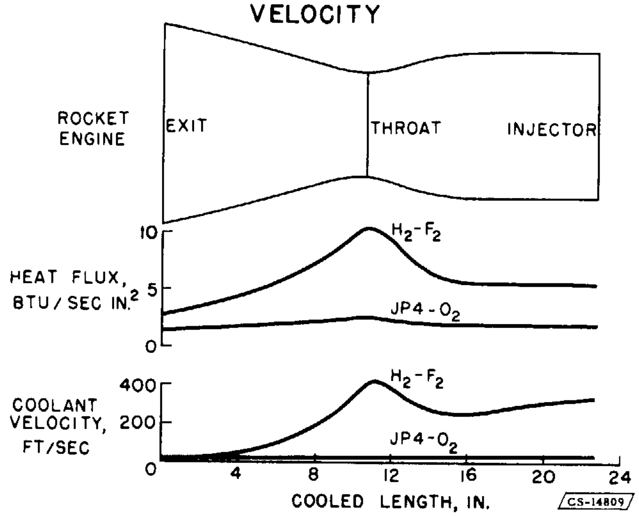

Figure 21

COOLANT TEMPERATURE AND PRESSURE IN A H_2-F_2 ENGINE

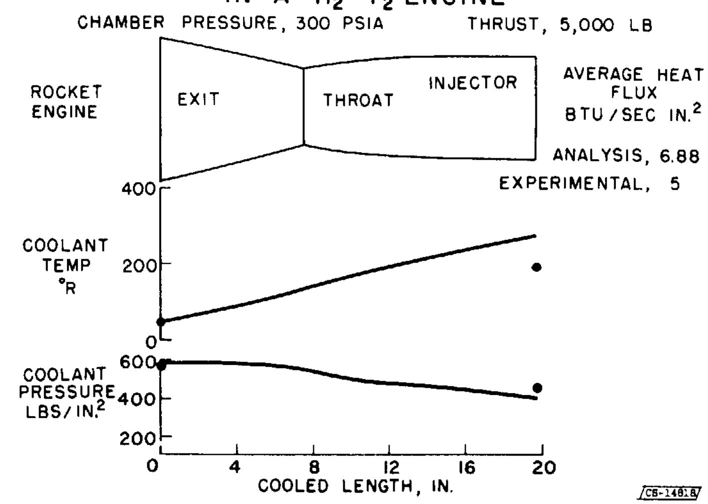

Figure 22

[redacted]

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